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RESEARCH MEMORANDUM

PRELIMINARY ATTEMPTS AT ISOTHERMAL COMPRESSION OF A SUPERSONIC AIR STREAM

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SUMMARY

Guided by analytical predictions, preliminary experiments were undertaken in an attempt to achieve isothermal (constant static temperature) compression of a supersonic air stream. Application of the process to a supersonic inlet diffuser at free-stream Mach numbers of 1.9 and 3.0 did not produce the theoretically predicted total-pressure rise. Large total-pressure losses due to momentum exchange between the inlet air stream and the coolant occurred, as expected, but the compensating rise in pressure theoretically associated with the available evaporation cooling was not observed. Tests at a Mach number of 3.0 with a heated air stream and multipoint upstream injection suggest that some gain in diffuser pressure recovery might be obtained with a full-scale inlet at the high stagnation temperature of supersonic flight.

INTRODUCTION

It is an established thermodynamic principle that an isothermal compression process is more efficient than the generally employed adiabatic compression process. In practical application of this theoretical principle to turbojet operation, steady-flow compressor efficiency was improved by means of water or water-alcohol injection during compression (ref. 1). Theoretical analysis (appendix A) indicates that a similar gain in total pressure may be achieved during diffusion of a supersonic stream if, instead of the conventional adiabatic (constant total temperature) compression process, isothermal (constant static temperature) compression be employed. In addition to improving engine cycle efficiency by raising the inlet total pressure, maintenance of constant stream static temperature relieves the problem of excessive compressor-inlet temperatures encountered by turbojet engines at supersonic speeds and increases the air-flow capacity of an engine of a given size.

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Application of the principle of isothermal compression to the diffusion and compression of a supersonic stream is by no means simple. It requires the rapid cooling of a supersonic stream as it is being decelerated without the introduction of excessive pressure losses. One method of achieving this desired cooling may be by the injection of an effective liquid coolant into the supersonic stream ahead of the air inlet. It is recognized (appendix B) that there may be a large pressure loss caused by injecting a liquid into a supersonic stream. However, if sufficient cooling can be achieved by evaporation of the coolant in the short time interval available, the gain in total pressure due to improved compression efficiency may more than compensate for the loss in total pressure caused by coolant injection.

A brief analytical study and a series of experiments were undertaken to evaluate the problems of cooling a supersonic air stream by liquid injection and to estimate the likelihood of approaching an isothermal compression process. Preliminary tests were conducted on injection techniques and an evaluation made of the evaporation achieved. The results were applied to axially symmetrical nose inlets, which were tested at Mach numbers 1.9 and 3.0. Nitrogen and ammonia were employed as coolants.

The symbols used are defined in appendix C.

APPARATUS AND PROCEDURE

Injection Tests

Before attempting to apply the isothermal-compression principle to a supersonic inlet, some preliminary experiments were undertaken to develop pumping and handling techniques for the liquid coolant. An additional purpose of these preliminary tests was to study briefly the injection of a liquid into a supersonic air stream. Specific objectives were to evaluate visually the penetration and the evaporation of the coolant, and to determine the effect of normal-shock proximity on coolant dispersion.

The tests were conducted in a 4- by 10-inch Mach number 2.0 wind tunnel. One wall of the tunnel was of optical-quality glass through which photographs could be taken. Total temperature in the test section was $95^\circ \pm 15^\circ$ F; total pressure was maintained at approximately 3300 pounds per square foot absolute.

As a safety precaution, the coolant employed in the initial tests was liquid nitrogen. For practical application of this principle, it is intended that a fuel, such as liquid methane, be utilized. The liquid nitrogen was stored at atmospheric pressure in a Dewar flask and was transferred to a vacuum-jacketed pressure vessel before use. It was pumped from there to the injector by introducing gaseous nitrogen, throttled to provide the desired coolant-flow rate, into the top of the pressure vessel. Flow rate was determined from the change in weight of the pressure vessel per unit time. A diagram of the installation is shown in figure 1.

Two types of injection systems were investigated. One was a flush-type commercial nozzle producing a fan-shaped spray, having a rated flow of 0.51 gallon of water per minute at a pressure of 60 pounds per square inch. The other consisted of a length of 5/16-inch brass tubing inserted in the stream, with two number 66 drill orifices opposite each other at right angles to the air stream.

The nozzle installation is shown in figure 2. The commercial nozzle was mounted flush with the tunnel wall, and the simple-orifice injector was placed along the tunnel centerline. A thermocouple probe was located approximately 1/2 foot downstream of the injection station.

Table I lists physical properties of the liquids used in this investigation and several others that may also be considered for this application.

Inlet Tests; Mach Number, 1.9

With the experience gained from the preliminary injection experiments, an effort was made to apply cooling to the air flowing into an actual supersonic inlet. Experiments were conducted in an 18- by 18-inch Mach number 1.9 wind tunnel at a simulated pressure altitude of 45,000 feet. Test-section total temperature was $150^{\circ}\pm10^{\circ}$ F. The diffuser model used was an isentropic spike inlet designed for operation at at Mach number of 3.0 (fig. 3, inlet III_a of ref. 2).

Both liquid nitrogen and liquid ammonia were investigated as coolants. The nitrogen, stored in a vacuum-walled pressure vessel, was pumped with gaseous nitrogen to discharge pressures of 200 pounds per square inch gage in the same manner as during the preliminary injector studies. The ammonia, stored in a pressure cylinder at ambient temperature, was pumped by its own vapor pressure of approximately 80 pounds per square inch.

Total-pressure recovery of the inlet was obtained from a survey of the total pressures at the diffuser exit. Inlet air flow was varied with a throttling valve at the diffuser exit, and total mass flow through the inlet was measured with a standard A.S.M.E. orifice. Coolant storage containers were mounted on weighing scales, from which coolant-flow rate was determined. The assumption that all the coolant entering the inlet was vaporized permits diffuser-exit temperature to be used as an independent check on the measured coolant-to-inlet-air ratio.

Four of the following five configurations investigated are sketched in figure 3:

(1) A 0.052-inch-diameter orifice at the spike tip

(2) A 0.010-inch annular slot located 1.25 inches from the spike tip and directed rearward at a 45° angle 3792

(3) A 0.010-inch annular slot located 1.25 inches from the spike tip and directed rearward behind a step in the spike contour and parallel to the spike surface

(4) A 0.090-inch-inside-diameter tube pointed upstream, discharging coolant 7.5 inches ahead of the cowl lip. (The tube was located on the spike centerline and was supported by a strut 10 inches ahead of the cowl lip and offset 0.5 inch from the inlet centerline.)

(5) (Not sketched) A 1/16-inch-diameter O-ring rolled into the spike, 1.25 inches from the tip

Inlet Tests; Mach Number, 3.0

To more fully explore the effectiveness of cooling the air stream some distance ahead of a supersonic inlet, additional experiments were conducted in an 18- by 18-inch Mach number 3.0 wind tunnel at a simulated pressure altitude of 82,000 feet. A ring injector having a double-wedge airfoil section was positioned alternatively at $1\frac{1}{2}$ and at 3 feet upstream of the inlet. Coolant was discharged through forty 0.020-inch-diameter orifices. The inlet employed was a 44° single-cone configuration (inlet I_a of ref. 2). Details of the installation are given in figure 4. Provision was made to raise the test-section stagnation temperature from 50°±10° F to 600°±50° F (ref. 3) in an attempt to improve the coolant heat transfer and evaporation rate. Both ammonia and water were investigated as coolants.

RESULTS AND DISCUSSION

Injection Tests

The flow pattern issuing from the orifice injector (fig. 2) is typical of that observed during these tests. The flow rate is 0.3 pound per minute at a pressure of 180 pounds per square inch. Jet penetration at right angles to the supersonic air stream varied between 1/2 and 1
3792 inch, depending upon the supply pressure; evaporation appeared to be complete approximately 3 inches downstream of the injector. The indicated temperature in a direct line downstream of the injector was -47° F. (Nitrogen injection temperature was -320° F and the test-section total temperature was 95° F.) An indication of the small degree of penetration achieved occurs from the increase in temperature from -47° to -27° F when the thermocouple was displaced 1/2 inch from its position directly downstream of the injector.

The general observations made for the simple orifice injector also applied for the commercial nozzles. In general, a very low temperature liquid can be injected into a supersonic air stream without freezing or clogging the injector. Although the nitrogen appeared to evaporate in a very short time interval and in only inches downstream of the point of injection, very little penetration into the air stream was observed.

Inlet Tests; Mach Number, 1.9

Following the initial exploratory tests on coolant injection, tests on the axially symmetrical inlet modified for coolant injection at the spike tip were undertaken. Inlet performance without coolant injection was first measured to provide a basis for comparison. The configuration tested without coolant flow is shown in figure 3, and the results obtained are plotted in figure 5. Peak pressure recovery was 0.94. A typical schlieren photograph near the peak-pressure-recovery condition is shown in figure 6(a).

The initial run with coolant flow was made with this same configuration. Liquid nitrogen was injected through the 0.052-inch-diameter orifice at the tip of the spike into the air entering the inlet. Inlet-pressure recovery for two coolant-flow rates is plotted in figure 5. In both cases, the pressure recovery as well as the peak mass-flow ratio fall below the no-coolant-flow values. Schlieren photographs (fig. 6(b)) showed a strong and abrupt disturbance at the spike tip, and it was observed that the magnitude of this disturbance increased with coolant-flow rate. Maximum pressure recoveries for the zero flow rate, as well as for the two coolant-flow rates investigated, are plotted in figure 7 as a function of the diffuser-exit temperature. The data clearly indicate that although the temperature of the air entering the diffuser was lowered, the greater the reduction in temperature, the greater the loss in total pressure.

In an effort to reduce the flow disturbance caused by injection at the spike tip, an annular gap injector, configuration 2 (fig. 3(a)) was employed. The results with this configuration are also given in figure 7, and although some improvement over injection at the spike tip may be noted, the pressure recovery remained considerably below the no-coolant-flow case. Schlieren photographs for a low and for a high coolant-flow rate are shown in figures 8(a) and (b), respectively. Again a strong and abrupt disturbance in the air flow was observed in the region of injection and the magnitude of the disturbance increased with coolant flow.

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To determine if the reduction in pressure recovery with coolant flow was due primarily to the flow disturbances and the resulting change in inlet-shock structure caused by the liquid injection, a 1/16-inch-diameter O-ring was rolled onto the spike and located at the annular gap of configuration 2. In schlieren photographs, the strength of the shock structure caused by the O-ring appeared similar to that caused with nitrogen injection through the annular orifice. However, as indicated in figure 7, maximum pressure recovery was only slightly less with the O-ring (0.9) than for no coolant flow. It appears, therefore, that shock losses alone do not account for the lower pressure recoveries observed with coolant flow.

A stepped spike designed to inject coolant along the spike surface, configuration 3 (fig. 3(b)) was next evaluated. This configuration was designed to reduce both the shock losses caused by coolant injection and the air and coolant mixing losses which, it was believed, accounted for the remainder of the total-pressure loss. A schlieren photograph of the flow about this spike with no coolant flow is shown in figure 9(a). Flow disturbances caused by the step introduced only a small drop in pressure recovery. A peak value of 0.9 was observed with no coolant flow. As anticipated, coolant flow tended to fill the region behind the step, apparently minimizing the surface discontinuity (fig. 9(b)). Again, peak pressure recovery decreased as coolant flow increased; although the coolant reduced diffuser-exit temperature as much as 200° F, pressure recovery (fig. 7) was considerably below the no-coolant-flow value.

These data indicate that at high coolant flows shock losses caused by the coolant or modifications to the inlet to allow coolant injection reduce total-pressure recovery much less than the reduction due to a momentum exchange between the air and the coolant. This indication is more readily apparent from the data of figure 10, which indicate the variation in peak pressure recovery as a function of coolant-to-inlet-air ratio. The dashed curve on figure 10 represents the computed pressure recovery without isothermal compression and accounts for the pressure loss due to the momentum exchange in a constant-area duct between the inlet-air stream and the liquid coolant. The exchange was assumed to occur at the local air velocity behind the oblique shock generated by the spike tip. A constant pressure loss of 10 percent, based on the no-injection inlet recovery of 0.9, has been assumed at all coolant-to-inlet-air ratios.

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Also included in figure 10 is the computed pressure recovery, including the momentum pressure loss, for the coolant-inlet-air ratio at which isothermal compression at a local Mach number of 1.6 can theoretically be achieved using nitrogen as a coolant. The data clearly fall along the calculated curve and away from the theoretical isothermal point, showing that the total-pressure loss accompanying coolant injection is due principally to acceleration of the coolant to local air velocity. Practically no effect of cooling on the pressure recovery was observed.

Liquid nitrogen had been selected as the coolant in the preliminary experiments because of its safety and ease of handling. Calculations had indicated (fig. 10) that for the ideal isothermal process a nitrogen-inlet-air ratio of 0.195 would be required, and little, if any, gain in total pressure would result. To reduce the coolant-inlet-air ratio needed for a given amount of cooling, some runs were made with configuration 3 using liquid ammonia, which has a latent heat of vaporization of 589.3 Btu per pound as compared with 86 for nitrogen. With ammonia as coolant, the calculated coolant-inlet-air ratio for the ideal isothermal process is reduced to 0.048.

Measured and calculated pressure recoveries as a function of ammonia-inlet-air ratio are given in figure 11. The calculations indicate that if isothermal compression could be accomplished with ammonia injection, a total-pressure recovery greater than 1.0 should result in spite of the usual inlet losses and the momentum exchange between the inlet air and the injected coolant. The experimental data, obtained with the step inlet (configuration 3), again indicated agreement with the calculated curve based on inlet losses and simple momentum exchange only, and the corresponding reduction in total-pressure recovery below the no-coolant-flow value resulted. Although the air at the diffuser exit was cooled to a value considerably below the ambient-air temperature, the pressure recovery gains of isothermal compression were not observed.

Because the results obtained indicated that the evaporation and cooling process lagged the momentum exchange and compression process, an attempt was made to reduce this lag. Coolant was sprayed into the supersonic air stream ahead of the step inlet (configuration 4, fig. 3(c)). The disturbances caused by the injector and its support, apparent from the schlieren photograph of figure 12(a), reduced the no-coolant-flow peak pressure recovery from a value of 0.9 to a value of 0.6. Injecting coolant caused a severe icing condition at the spike tip (fig. 12(b)), and reduced peak pressure recovery to 0.5. Since the amount of water in the tunnel air was approximately one part in a thousand, it appears that the ice was probably solid ammonia, which freezes at -107° F. The ice, permitted to build up to a thickness of approximately 1/8-inch, melted when the coolant flow was stopped.

Because little if any gain in pressure recovery was obtained from evaporation of the coolant, it must again be concluded that the cooling occurred primarily in the subsonic portion of the diffuser. To improve coolant distribution and thereby achieve more rapid evaporation, succeeding tests were run with a multipoint upstream injector. These tests were run at a stream Mach number of 3.0.

Inlet Tests; Mach Number, 3.0

Details of the multipoint injector and the 44° single-cone spike inlet used in these tests are given in figure 4. Preliminary attempts at checking out the injection system with water resulted in the formation of ice on the injector ring. With liquid ammonia as a coolant, no ice formed. 3792

Presence of the ring injector $1\frac{1}{2}$ feet ahead of the inlet appreciably reduced the stream total pressure available to the inlet, and lowered inlet total-pressure recovery. There appeared to be some evidence that this total-pressure loss could be reduced slightly by increasing the distance between the injector and the inlet to 3 feet.

Results observed with liquid ammonia are summarized in figure 13. The mass-flow ratio is based on the undisturbed stream maximum inlet mass flow. The results of coolant injection into the air stream well ahead of the inlet observed at a Mach number of 1.9 were confirmed at a Mach number of 3.0. In spite of improved coolant distribution, diffuser critical total-pressure recovery fell from a value of 0.54 with the air stream unobstructed to 0.40 at a coolant-air ratio of 0.024.

More encouraging results were obtained at a higher stream total temperature in an attempt to increase coolant evaporation and heat transfer to the air stream. A core of hot air with a diameter greater than that of the diffuser lip was produced within the tunnel test section by the technique described in reference 3. Whereas the previous experiments were made at a stream total temperature of approximately 50° F, the core was maintained at approximately 600° F, a realistic stagnation temperature for flight at a Mach number of 3.0. The effect of raising the core temperature was to reduce the test section (core) Mach number and introduce some total-pressure losses. Consequently, the inlet total-pressure recovery with the injector in place but no coolant flow was 3 percentage points less when the core was heated. Only a small additional drop in pressure recovery resulted upon the injection of ammonia (liquid-inlet-air ratio, 0.021). It may therefore be concluded that at a temperature near that expected in flight at a Mach number of 3, the more effective cooling achieved compensated to a large degree for the momentum losses introduced by coolant injection. Even better pressure recovery was observed when the injector spacing was changed from $1\frac{1}{2}$ to 3 feet. These results suggest that the isothermal compression process might be successfully applied to a full-scale inlet configuration at the high stagnation temperature of supersonic flight.

It was recognized that application of the isothermal cooling process must be accompanied by a specific variation in flow area. However,
3792 contraction of the streamlines ahead of the inlet lip caused by the upstream coolant injection preclude a theoretical estimate of the possible pressure gain based solely on the actual contraction ratio of this specific inlet configuration.

SUMMARY OF RESULTS

Preliminary attempts at raising inlet pressure recovery by cooling the supersonic air stream entering an inlet diffuser while this stream is being decelerated and compressed by shock waves generated by the inlet were only partially successful. The increase in total pressure theoretically predicted by cooling the air stream during compression was not achieved. Instead, large total-pressure losses were introduced by liquid-coolant injection. These pressure losses were caused by the momentum exchange between the air stream and the coolant due to the required acceleration of the coolant to local stream velocity.

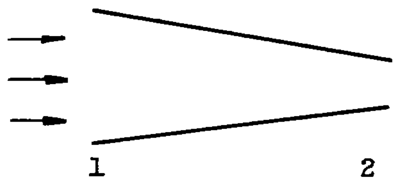
The experiments demonstrated, however, that evaporative cooling becomes more effective as the air-stream stagnation temperature and the distance of the cooling injection point ahead of the inlet are increased. These results suggest that the effectiveness of liquid injection ahead of an inlet should be reevaluated on large-scale inlets in air streams with high stagnation temperatures.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, October 4, 1955

APPENDIX A

DERIVATION OF ISOTHERMAL FLOW RELATIONS

Consider the one-dimensional flow, either subsonic or supersonic, of an ideal gas in a nonconstant-area duct (see following sketch).



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If the flow between stations 1 and 2 is defined as isothermal (constant static temperature, $t_2 = t_1$), then the equation of state can be written as

$$p_1 v_1 = p_2 v_2 = \text{constant} \tag{A1}$$

The conservation of mass between the two stations states that

$$\rho_1 A_1 V_1 = \rho_2 A_2 V_2 \tag{A2}$$

and the conservation of energy that

$$c_p T_1 = c_p T_2 - Q \tag{A3}$$

In more general form, the energy equation is also given by

$$\frac{V_1^2}{2g} + c_p t_1 + W = \frac{V_2^2}{2g} + c_p t_2 - Q \tag{A4}$$

Also, for the general isothermal process, any work W put into the system is given by

$$W = \frac{V_2^2 - V_1^2}{2g} + R t_1 \log_e \frac{p_2}{p_1} \tag{A5}$$

(See ref. 4.)

Equation (A2) can be reduced to give

$$\frac{p_2}{p_1} = \frac{A_1 M_1}{A_2 M_2} \tag{A6}$$

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Combination of the defining relations between total and static temperatures at stations 1 and 2 in terms of Mach number and γ produces the following expression for total-temperature ratio:

$$\frac{T_2}{T_1} = \frac{1 + \frac{\gamma - 1}{2} M_2^2}{1 + \frac{\gamma - 1}{2} M_1^2} \tag{A7}$$

Further, combination of the defining relation between the static and total pressures in terms of M and γ with equations (A6) and (A7) gives the total-pressure ratio

$$\frac{P_2}{P_1} = \frac{A_1}{A_2} \frac{M_1}{M_2} \left(\frac{T_2}{T_1}\right)^{\frac{\gamma}{\gamma - 1}} \tag{A8}$$

Since for the case being considered, no work is done on the fluid, $W = 0$, and $t_1 = t_2$, equation (A4) reduces to

$$Q = \frac{V_2^2 - V_1^2}{2g} \tag{A9}$$

or

$$\frac{Q}{c_p t} = \frac{\gamma - 1}{2} M_1^2 \left[\left(\frac{M_2}{M_1}\right)^2 - 1\right] \tag{A10}$$

Also, equation (A5) becomes

$$\frac{V_1^2 - V_2^2}{2g} = Rt_1 \log_e \frac{p_2}{p_1} \tag{A11}$$

To obtain the relation between flow areas at stations 1 and 2, equation (A6) is substituted into equation (A11), giving

$$\frac{A_1}{A_2} = \frac{M_2}{M_1} e^{\frac{\gamma}{2} M_2^2 \left(\frac{M_1^2}{M_2^2} - 1\right)} \tag{A12}$$

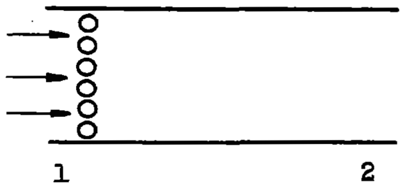
These expressions describe flow compression by means of isothermal compression of the stream. Solutions to equations (A6) and (A8) are plotted in figure 14(a), and equations (A7), (A10), and (A12) in figure 14(b). Although it was assumed that the flow is decelerated isothermally from its initial velocity to a final Mach number of unity, these charts can be utilized for any other final Mach numbers by the same methods employed with the conventional isentropic flow charts. 3792

A comparison between the usual compression process for the diffusion of a supersonic air stream and the isothermal process shows that total and static pressures after compression are much greater with the isothermal process. In fact, for the isothermal process total pressure is greater after diffusion than before. For example, in diffusing isothermally without a shock from $M_1 = 2.0$ to $M_2 = 1.0$, final total pressure $P_2 = 1.98P_1$. In the isentropic case, $P_2 = P_1$. For the isothermal case, the final static pressure $p_2 = 8.16p_1$, whereas for the isentropic case $p_2 = 4.13p_1$. Of course, to achieve the pressure recovery gains that isothermal theory suggests are possible, the heat extraction from the air stream must be accompanied by appropriate changes in stream-tube cross-sectional area. Consequently, the diffuser design would necessarily be different from that for the isentropic case.

APPENDIX B

PRESSURE LOSS FROM FLUID INJECTION INTO AN AIR STREAM

The process of injecting a liquid coolant into the air stream ahead of a supersonic inlet is visualized as occurring in a constant-area duct. The exchange of momentum between the inlet air and the coolant is taken to occur prior to the heat transfer from coolant to air. This momentum exchange is accompanied by a loss in stream total pressure that may be estimated by assuming the following one-dimensional system:



Liquid is injected as a spray at station 1 into an air stream flowing through a constant-area duct. It is assumed that the liquid is dispersed between stations 1 and 2, and is uniformly distributed when station 2 is reached. If it is further assumed that the liquid volume at both stations 1 and 2 is small compared with the gas volume, then

$$\rho_1 A_1 V_1 (1 + l/a) = \rho_2 A_2 V_2 (1 + l/a) \tag{B1}$$

Since $A_1 = A_2$ and T_1 is assumed equal to T_2, this equation becomes

$$\frac{p_1 V_1}{t_1} = \frac{p_2 V_2}{t_2} \tag{B2}$$

or

$$\frac{p_2}{p_1} = \frac{M_1}{M_2}\sqrt{\frac{t_2}{t_1}} \tag{B3}$$

In terms of total pressure, equation (B3) becomes

$$\frac{P_2}{P_1} = \frac{M_1}{M_2}\left(\frac{1 + \frac{\gamma - 1}{2} M_2^2}{1 + \frac{\gamma - 1}{2} M_1^2}\right)^{\frac{1}{2} - \frac{\gamma}{\gamma - 1}} \tag{B4}$$

To eliminate M_2 from equation (B4) and determine the effect of liquid-air ratio l/a on the relation between pressure loss and initial Mach number, the momentum relation between stations 1 and 2 is utilized. Thus,

$$p_1 A_1 + m_a V_1 + m_l V_l = p_2 A_2 + (m_a + m_l) V_2 \tag{B5}$$

or

$$\left(1 - \frac{p_2}{p_1}\right)\frac{A_1 p_1}{m_a} + V_1 + \frac{l}{a} V_l = \left(1 + \frac{l}{a}\right) V_2 \tag{B6}$$

Substituting $\rho_1 A_1 V_1$ for m_a, changes equation (B6), in terms of Mach number, to

$$\left(1 - \frac{p_2}{p_1}\right)\frac{1}{\gamma M_1} + M_1 + \frac{l}{a} M_l = \left(1 + \frac{l}{a}\right) M_2 \sqrt{\frac{t_2}{t_1}} \tag{B7}$$

where M_l is expressed in terms of the local speed of sound of the air. Substituting equation (B3) into equation (B7) produces

$$\left(1 - \frac{p_2}{p_1}\right)\frac{1}{\gamma M_1} + M_1 - M_2 \sqrt{\frac{t_2}{t_1}} = \frac{l}{a}\left(M_2 \sqrt{\frac{t_2}{t_1}} - M_l\right) \tag{B8}$$

Solving yields

$$\frac{l}{a} = \frac{\left(1 - \frac{p_2}{p_1}\right)\frac{1}{\gamma M_1} + M_1 - M_2 \sqrt{\frac{t_2}{t_1}}}{M_2 \sqrt{\frac{t_2}{t_1}} - M_l} \tag{B9}$$

The simultaneous solution of equations (B4) and (B9) yields the relation between M_1, l/a, and P_2/P_1 for a constant value of M_l. Results for M_l = -0.4, 0, and 0.4 are plotted in figure 15.

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APPENDIX C

SYMBOLS

The following symbols are used in this report:

A	area, sq ft
c_p	specific heat at constant pressure, ft-lb/lb/°F
g	acceleration due to gravity, ft/sec^2
l/a	liquid-to-inlet-air ratio, weight basis
M	Mach number
m	mass flow, slugs/sec
P	total pressure, lb/sq ft
p	static pressure, lb/sq ft
Q	heat energy, ft-lb/lb
R	gas constant, ft-lb/lb/°F
T	total temperature, °R
t	static temperature, °R
V	velocity, ft/sec
v	specific volume, cu ft/lb
W	work done on fluid, ft-lb/lb
γ	ratio of specific heats
ρ	density, slugs/cu ft

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Subscripts:

0	free stream
1	station 1
2	station 2

a air

l liquid

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TABLE I. - PROPERTIES OF COOLANTS

Coolant (all in liquid state)	Boiling point at atmospheric pressure, °F	Latent heat of vaporization, Btu/lb	Specific gravity	Lower heating value, Btu/lb	Stoichiometric fuel-air ratio
Nitrogen	-320.4	86	0.808	------	-----
Ammonia	-28	589.3	.712	------	-----
Water	212	970.3	1.000	------	-----
Methane	-258.5	248	.415	21,529	0.058
Gasoline	280 av.	116	.713	18,500	.067
Diborane	-134.5	124.5	.447	33,513	.067
Hydrogen	-422.9	194	.070	51,608	.029

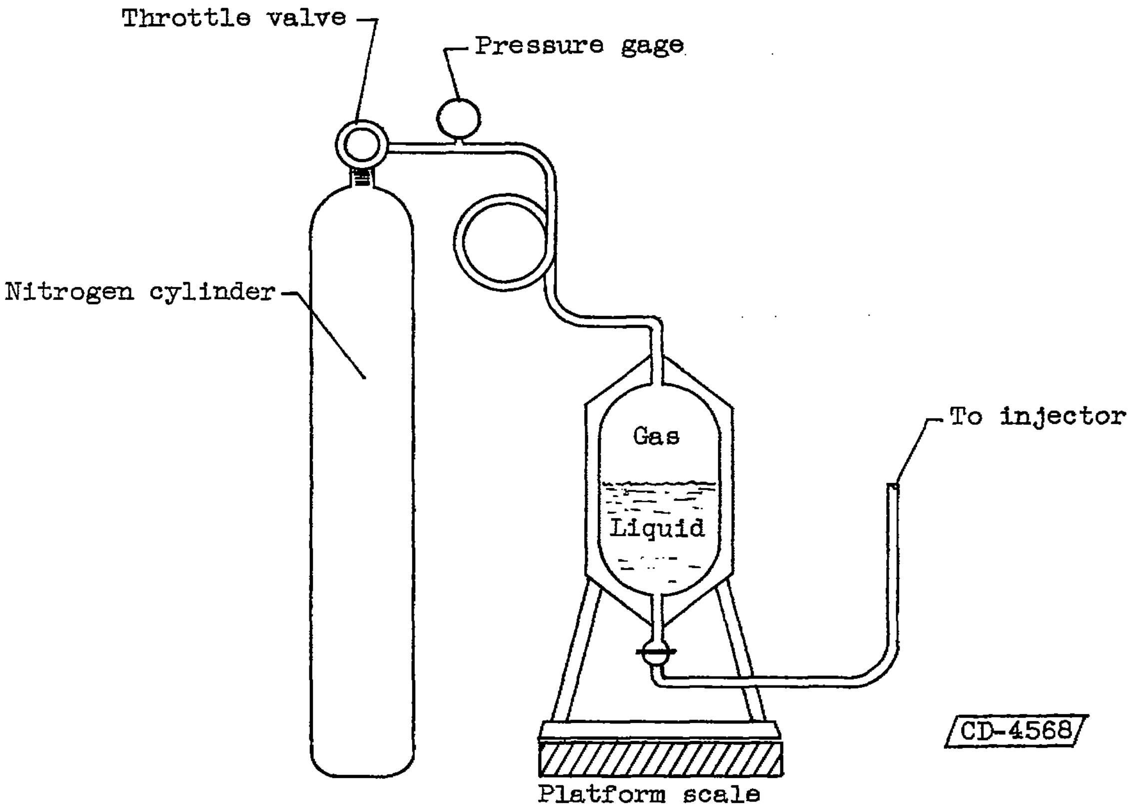


Figure 1. - Schematic diagram of nitrogen-injection system.

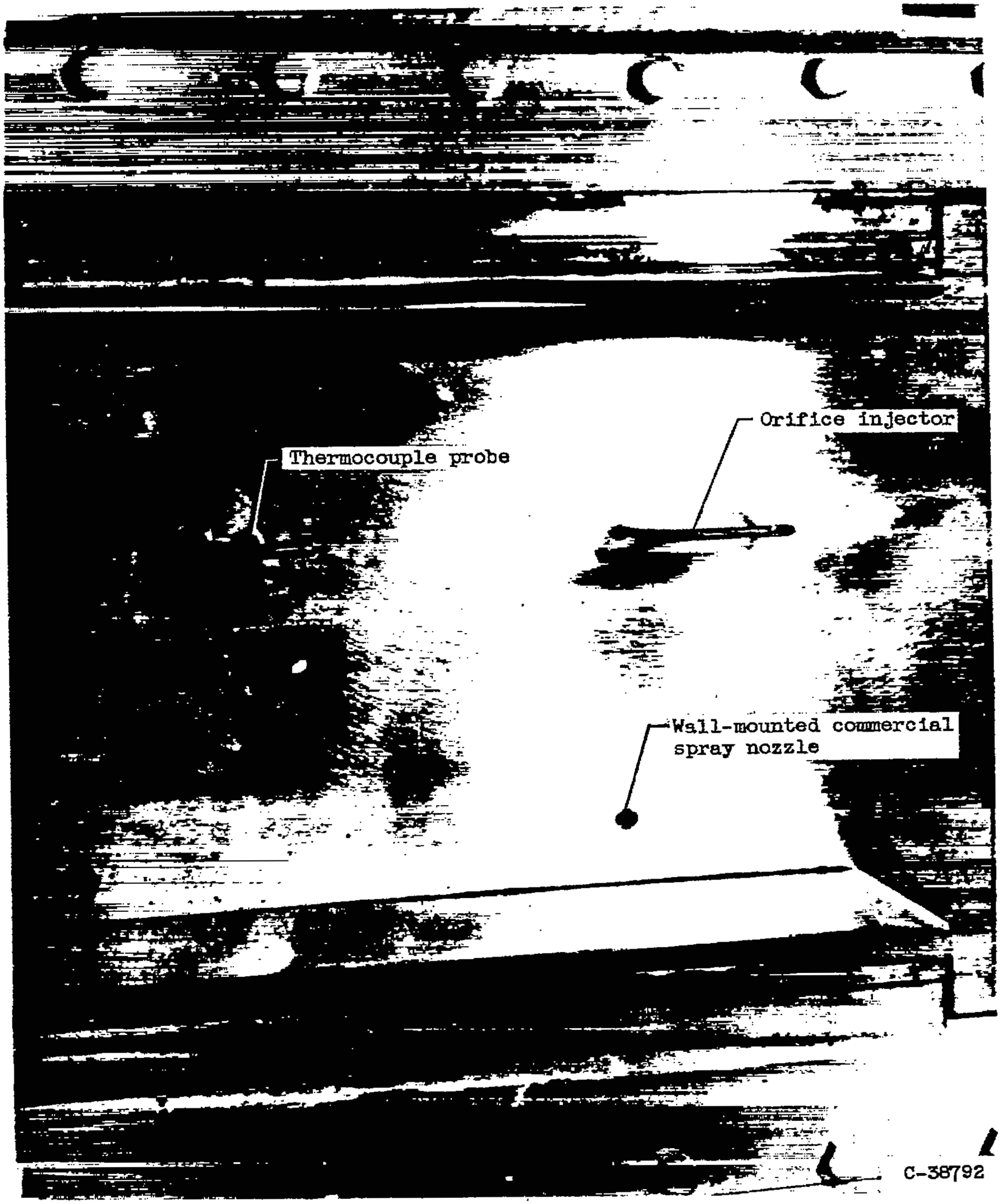
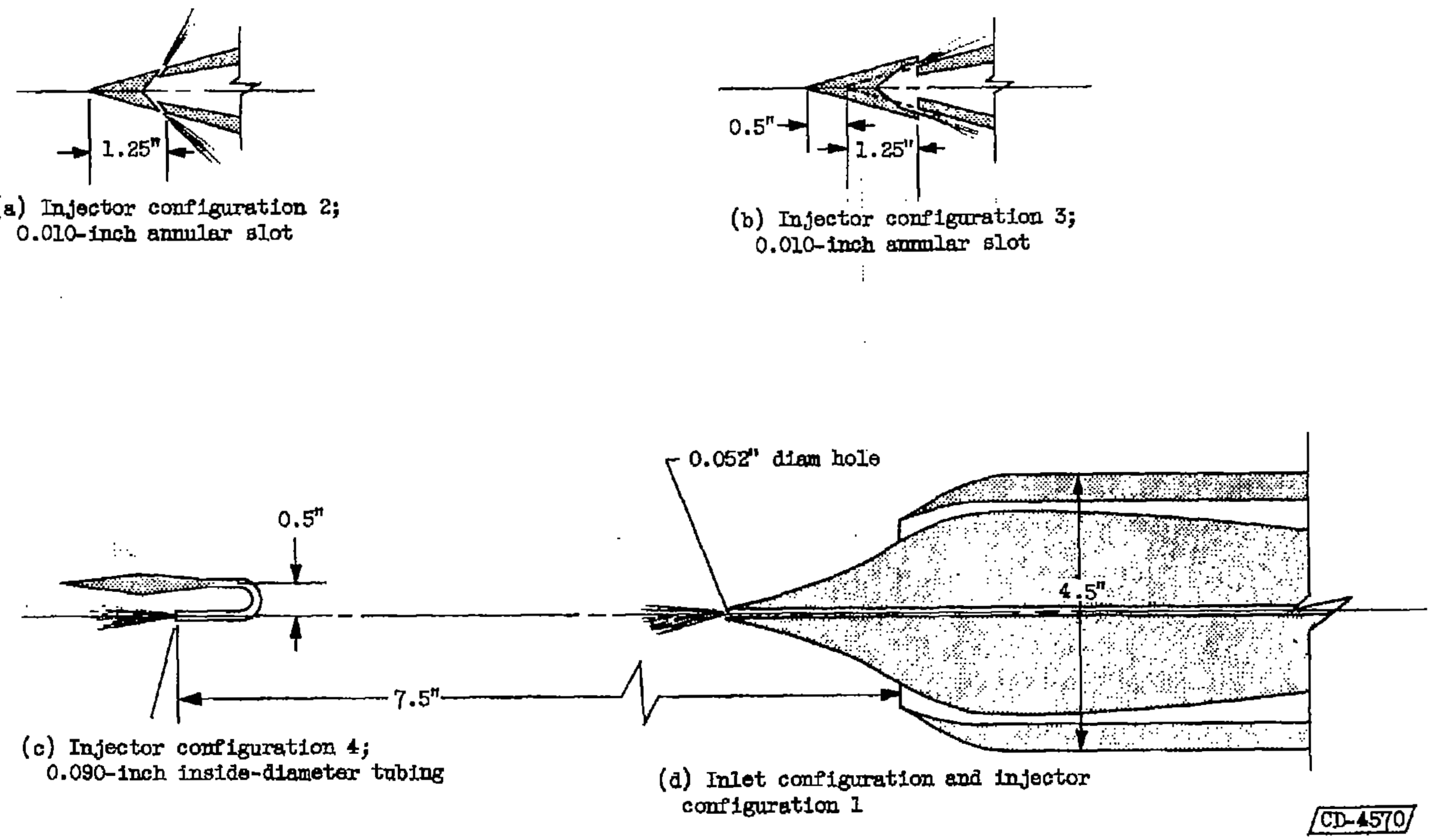


Figure 2. - Liquid-nitrogen injection from simple orifice injector in 4- by 10-inch tunnel. Free-stream Mach number, 2.0.



(a) Injector configuration 2; 0.010-inch annular slot

(b) Injector configuration 3; 0.010-inch annular slot

(c) Injector configuration 4; 0.090-inch inside-diameter tubing

(d) Inlet configuration and injector configuration 1

Figure 3. - Isentropic-spike and coolant-injector configurations.

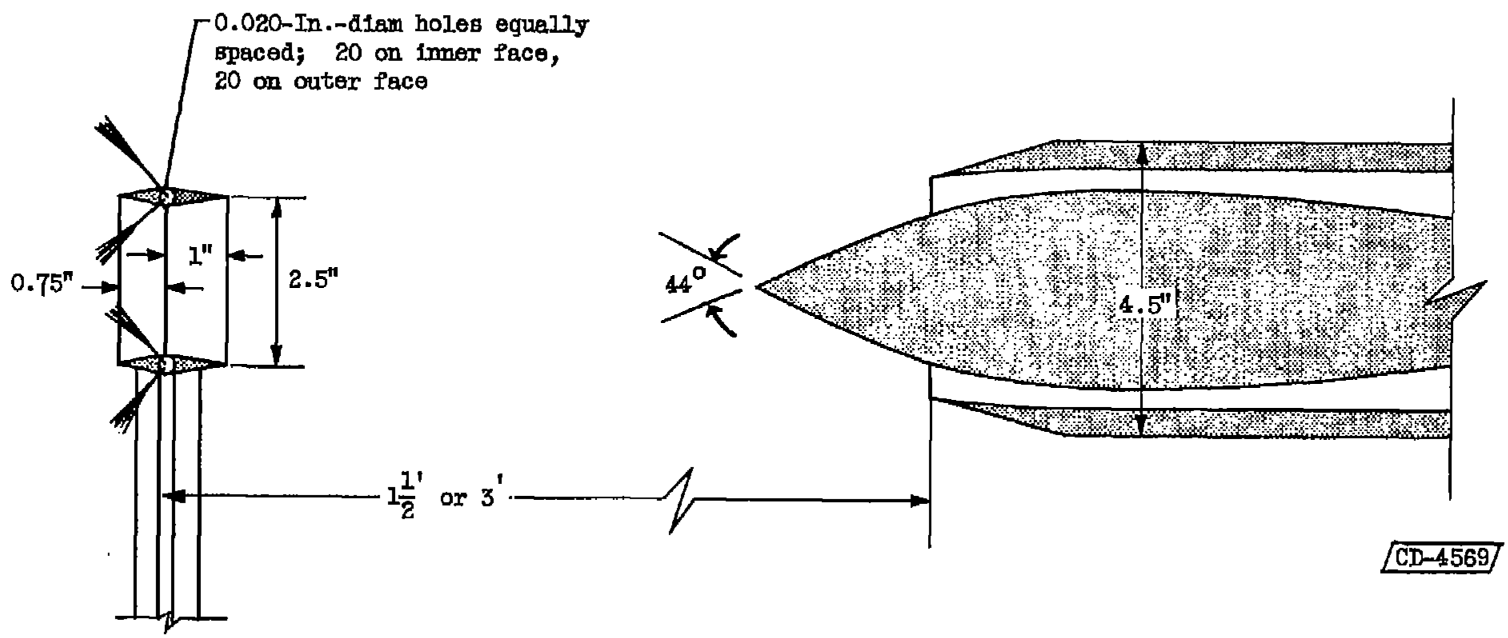


Figure 4. - Installation details of inlet and multipoint injector used at Mach number of 3.0.

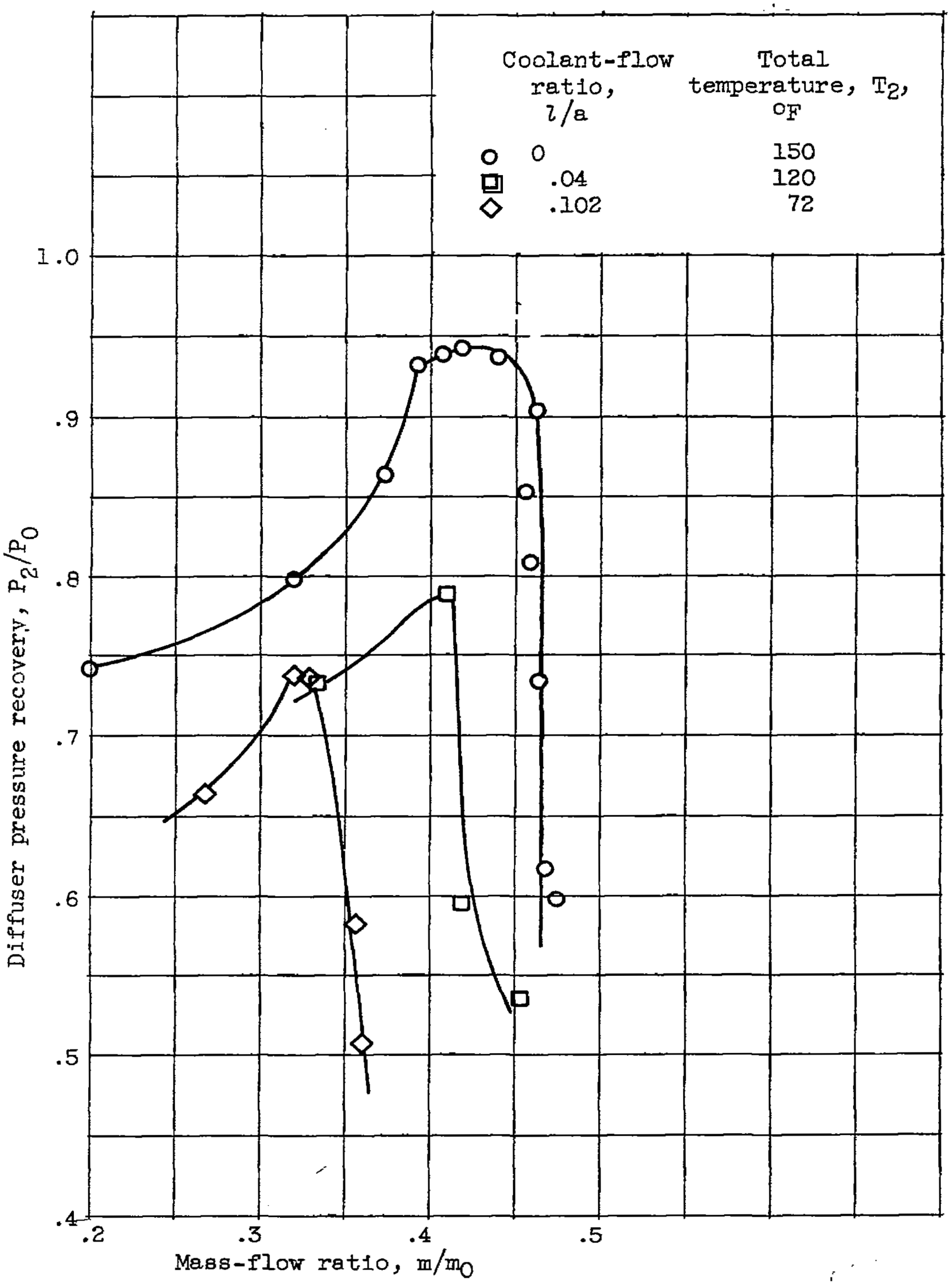


Figure 5. - Diffuser pressure-recovery characteristics. Injector configuration 1; free-stream total temperature, 150° F; free-stream Mach number, 1.9.

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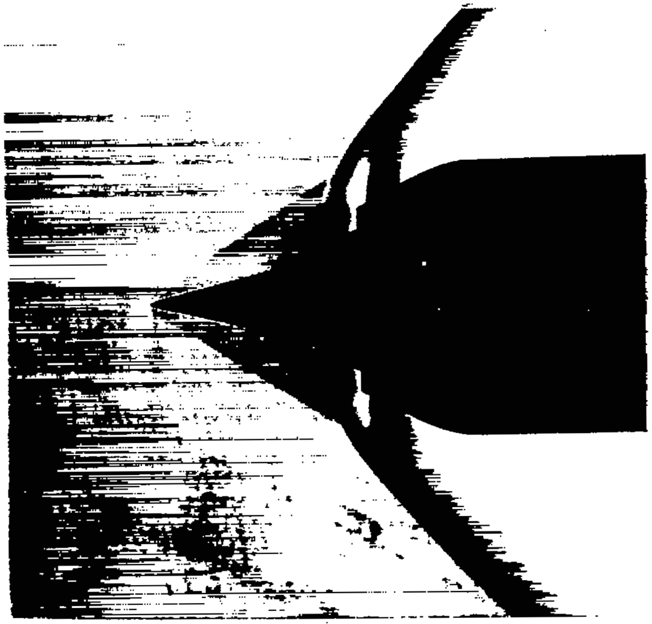
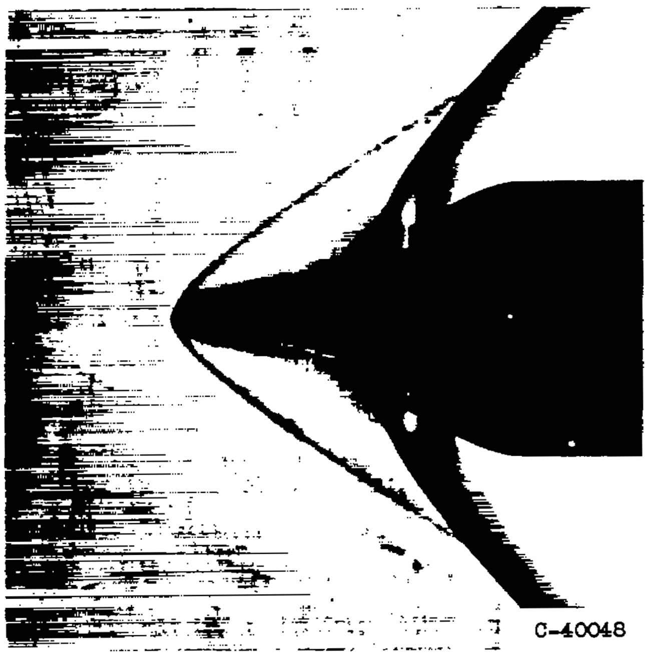


(a) Maximum air flow. No coolant injection.



(b) Nitrogen injected through spike tip.

Figure 6. - Schlieren photographs of configuration 1. Free-stream Mach number, 1.9.

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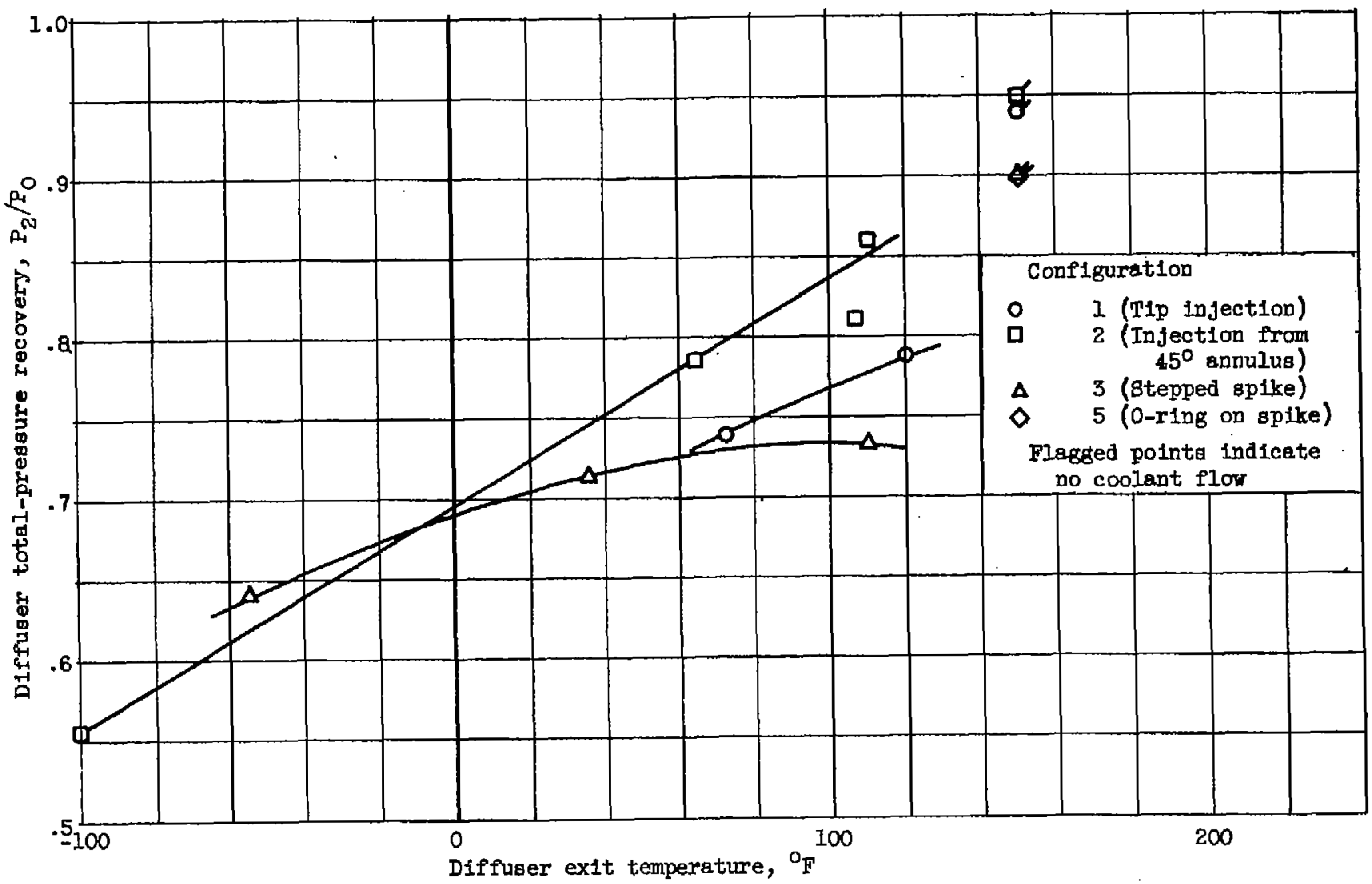
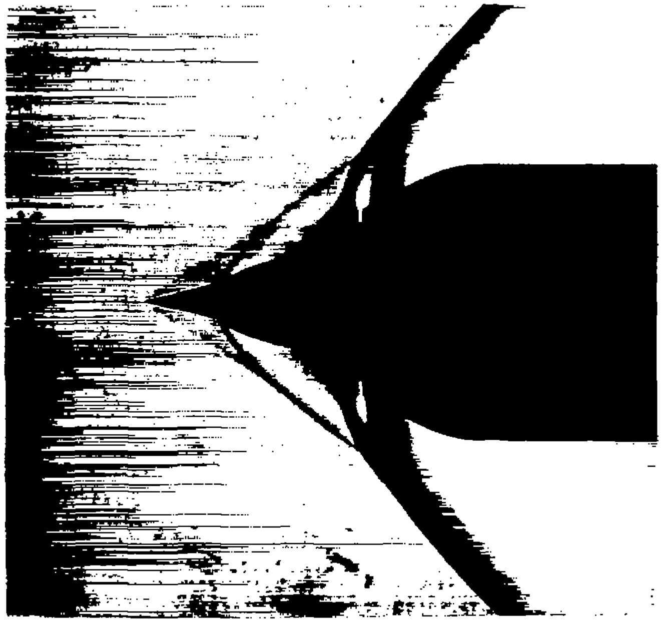
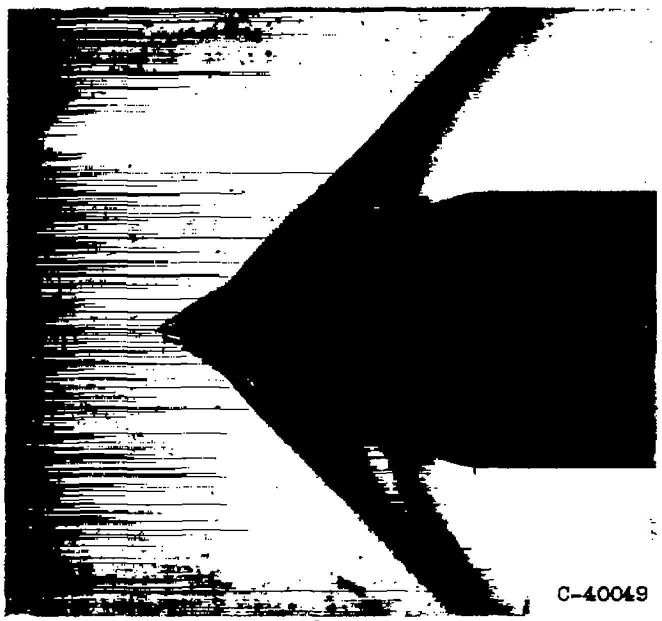


Figure 7. - Effect of nitrogen flow rate on maximum pressure recovery of configurations investigated. Free-stream total temperature, 150° F; free-stream Mach number, 1.9.

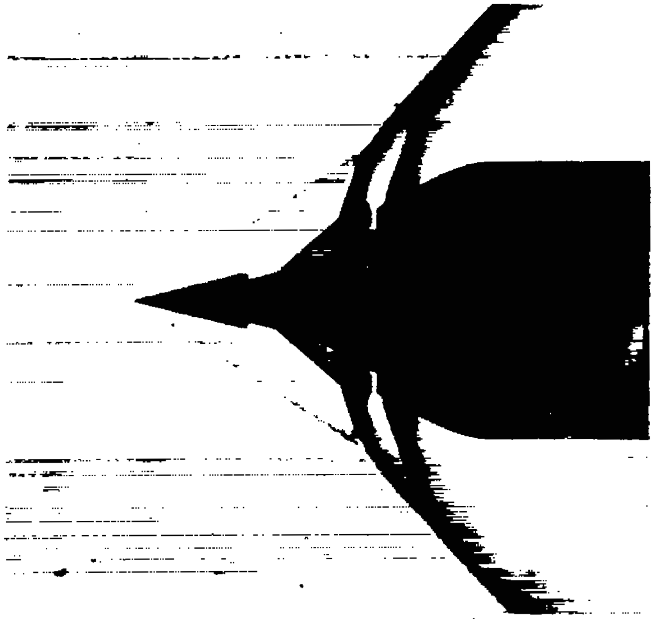


(a) Nitrogen injected at low flow rate.

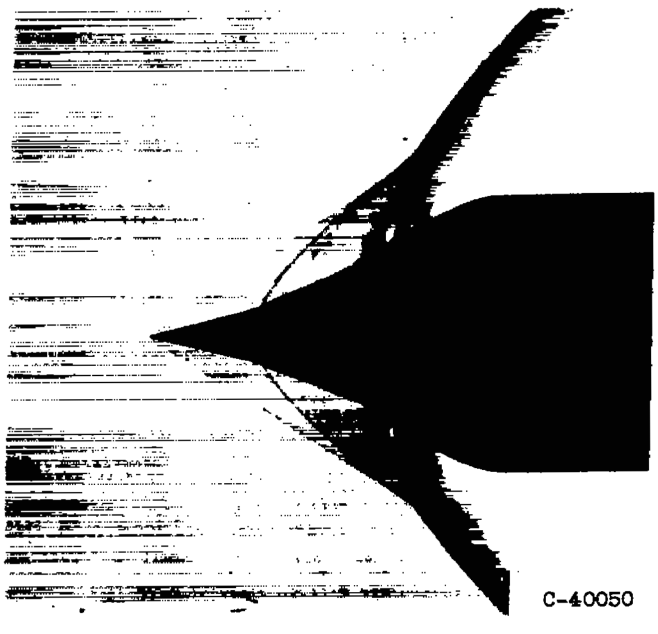


(b) Nitrogen injected at high flow rate.

Figure 8. - Schlieren photographs of configuration 2. Free-stream Mach number, 1.9.



(a) No coolant flow.



(b) Nitrogen injection behind step.

Figure 9. - Schlieren photographs of configuration 3. Free-stream Mach number, 1.9.

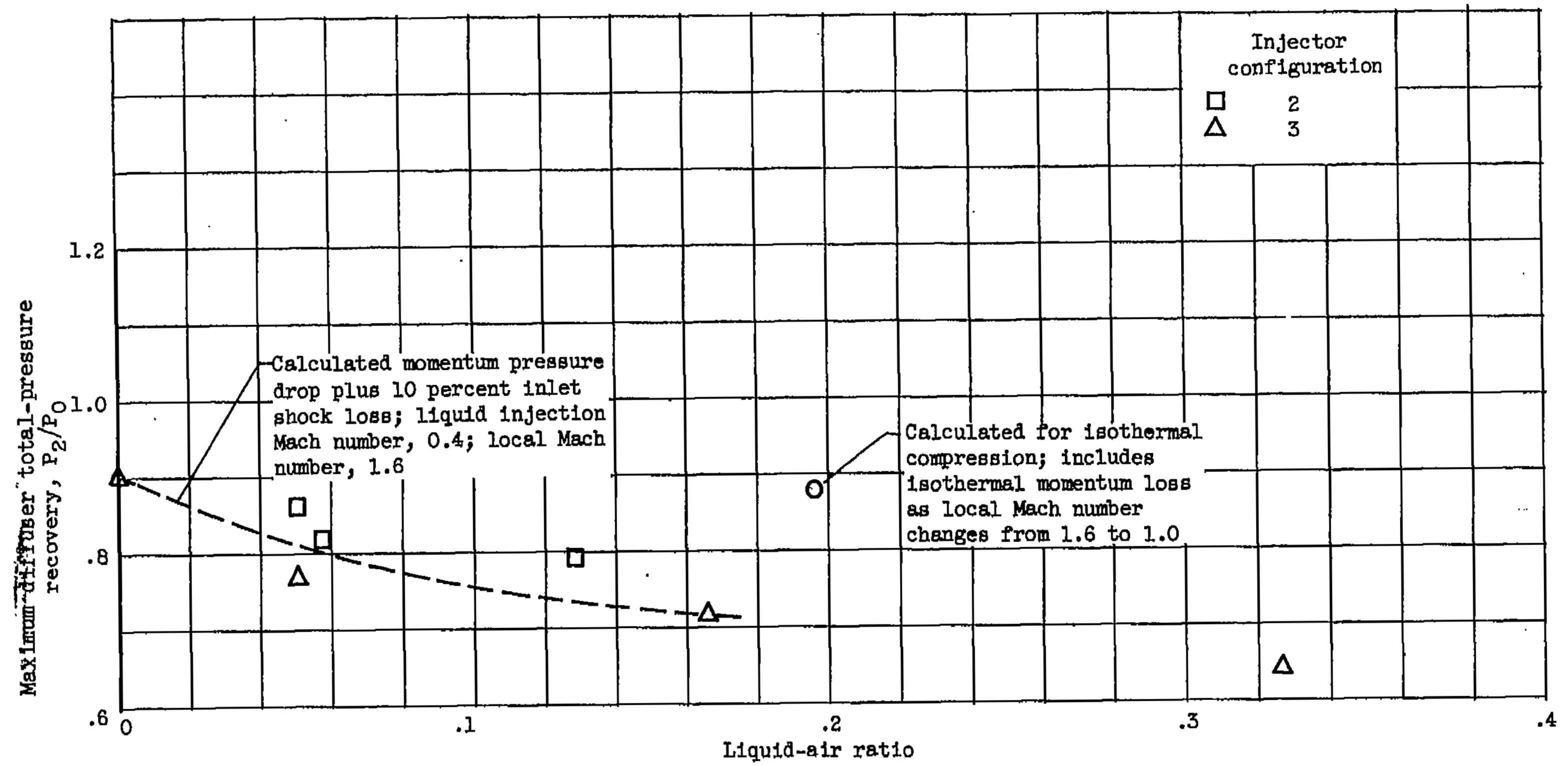


Figure 10. - Summary of inlet injection data with nitrogen. Free-stream Mach number, 1.9.

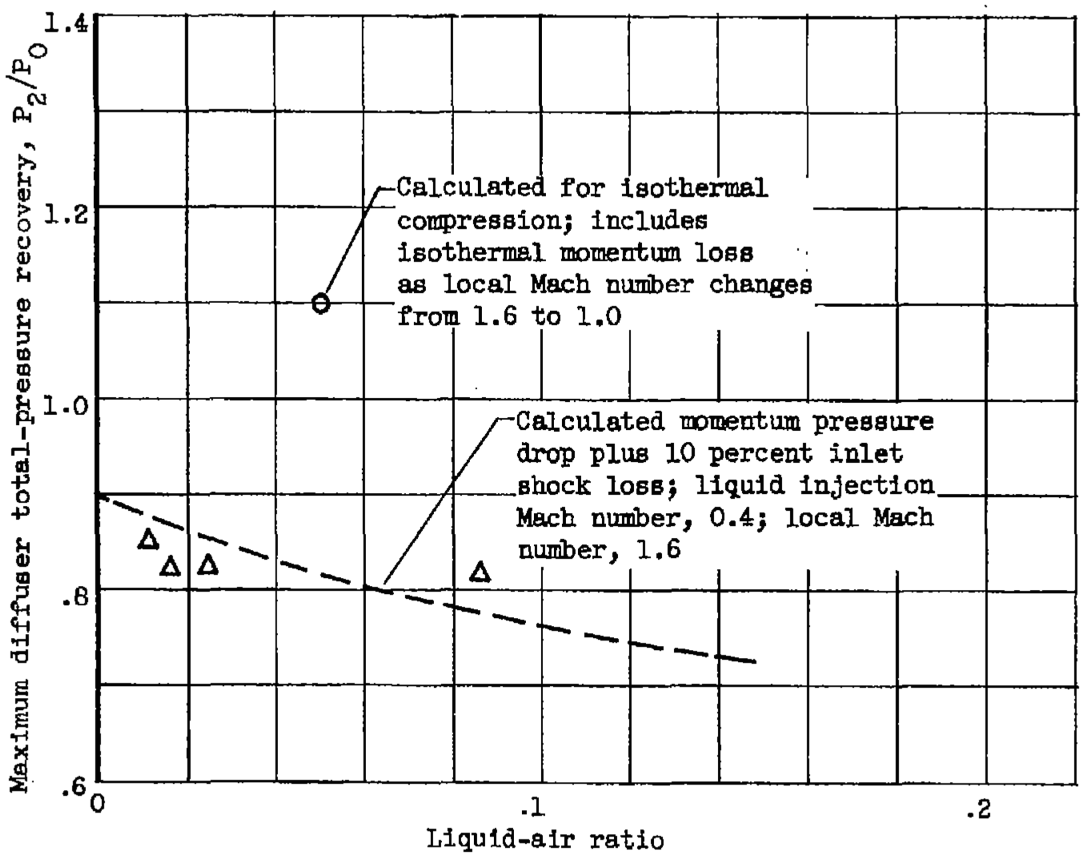


Figure 11. - Summary of inlet injection data with ammonia for injector configuration 3. Free-stream Mach number, 1.9.

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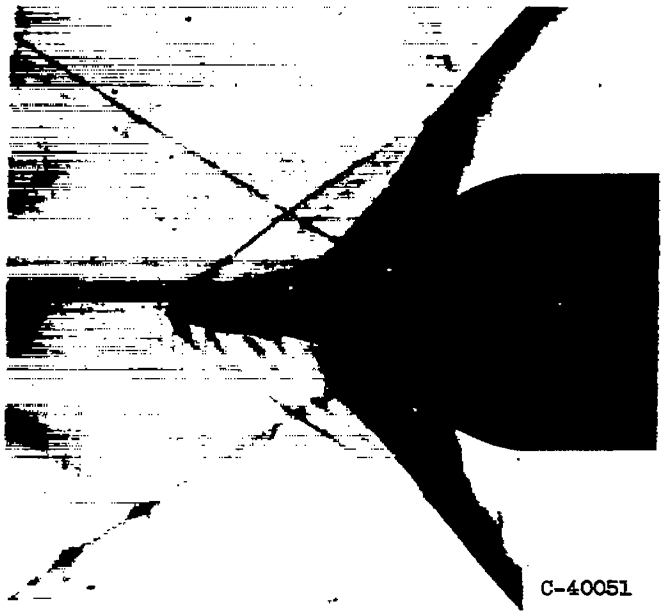
(a) No coolant injection.



(b) Solid ammonia formation on spike tip.

Figure 12. - Schlieren photographs of configuration 4. Free-stream Mach number, 1.9.

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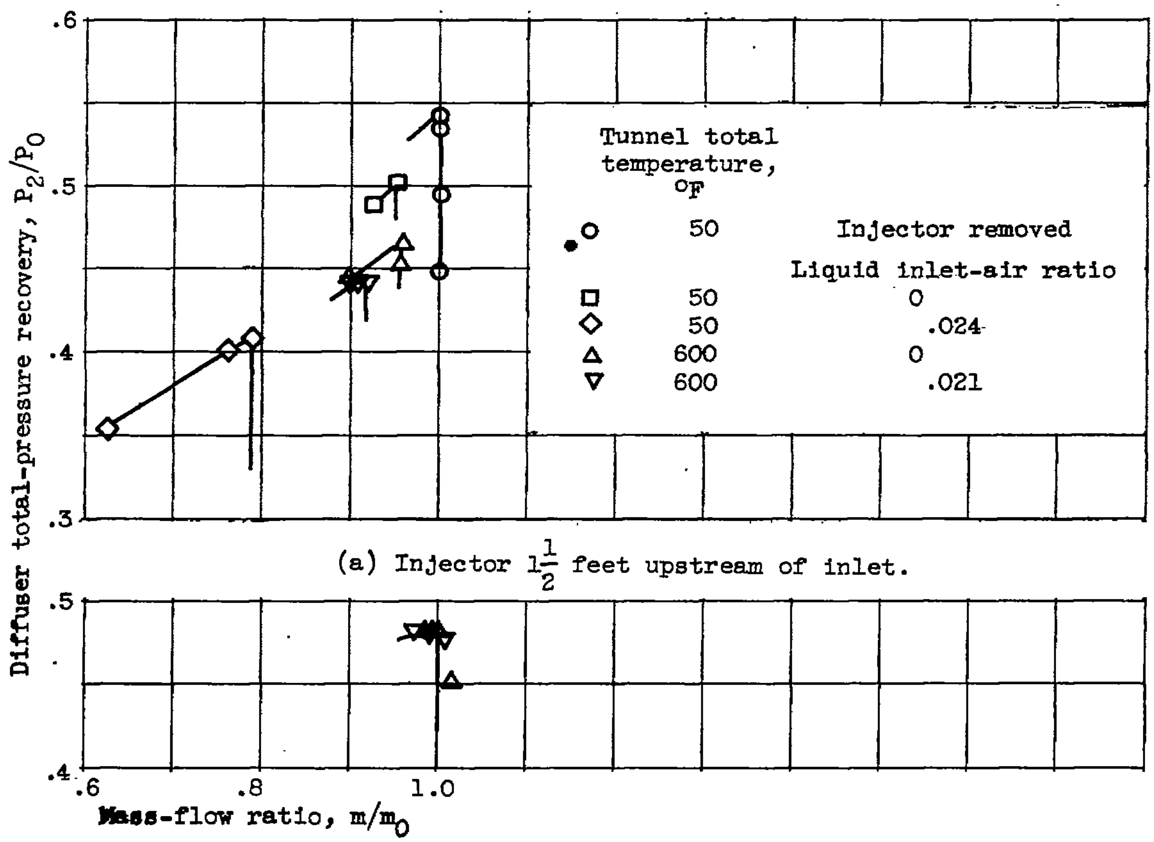

(a) Injector $1\frac{1}{2}$ feet upstream of inlet.

(b) Injector 3 feet upstream of inlet.

Figure 13. - Ammonia injection at free-stream Mach number, 3.0; multipoint injector.

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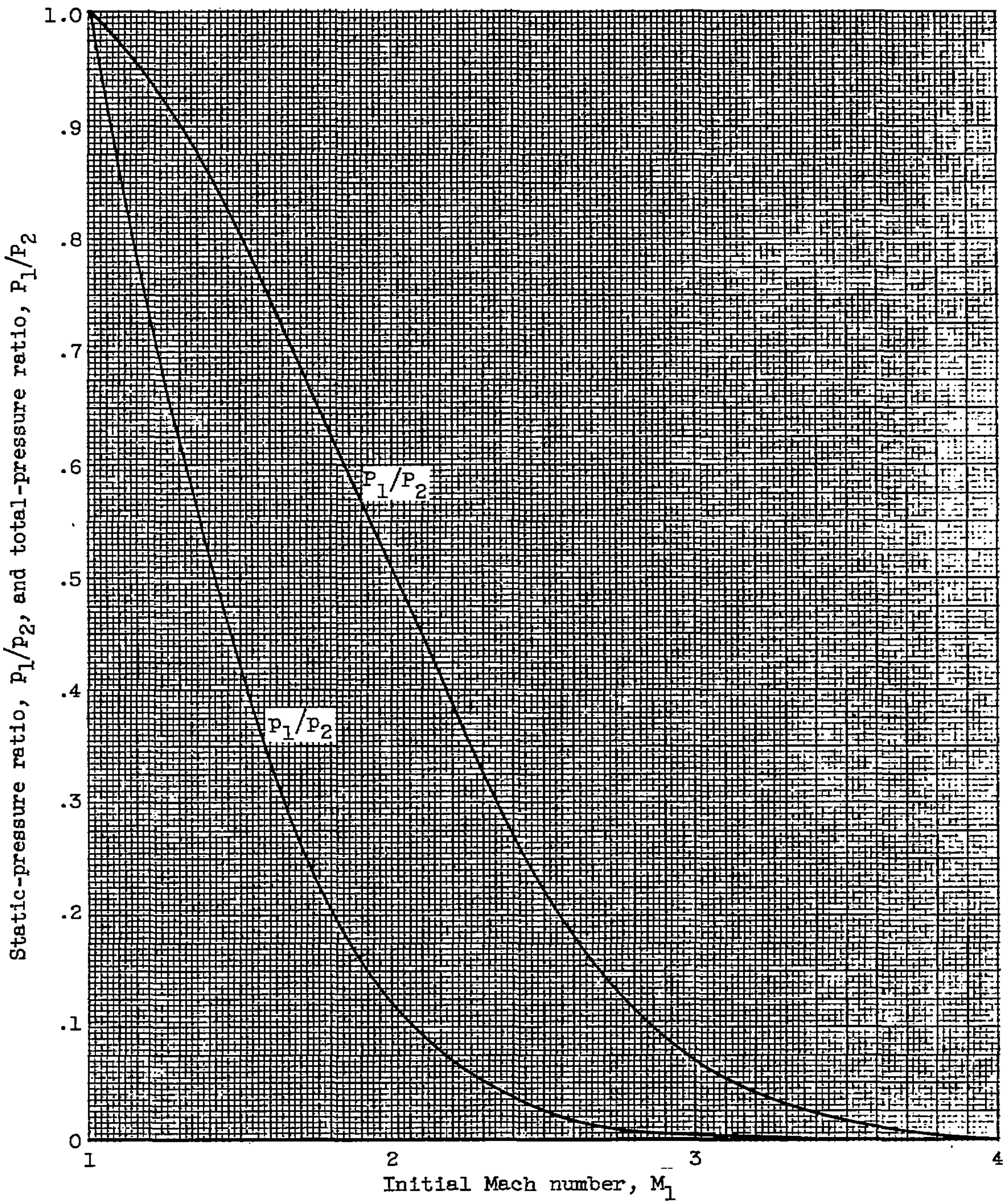

(a) Static- and total-pressure ratios.

Figure 14. - Isothermal flow relations. Final Mach number, 1.

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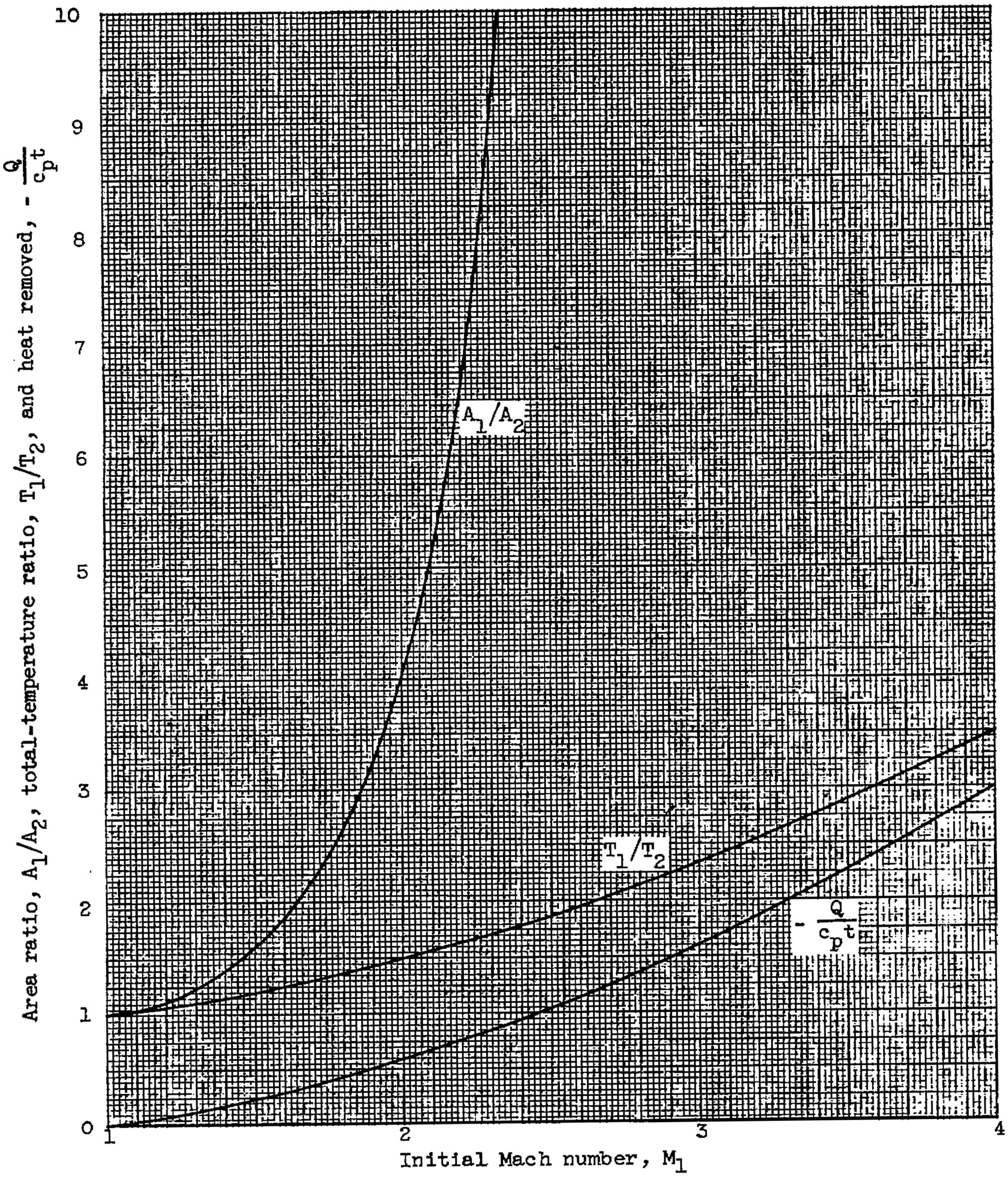

(b) Area and total-temperature ratios and heat removed.

Figure 14. - Concluded. Isothermal flow relations. Final Mach number, 1.

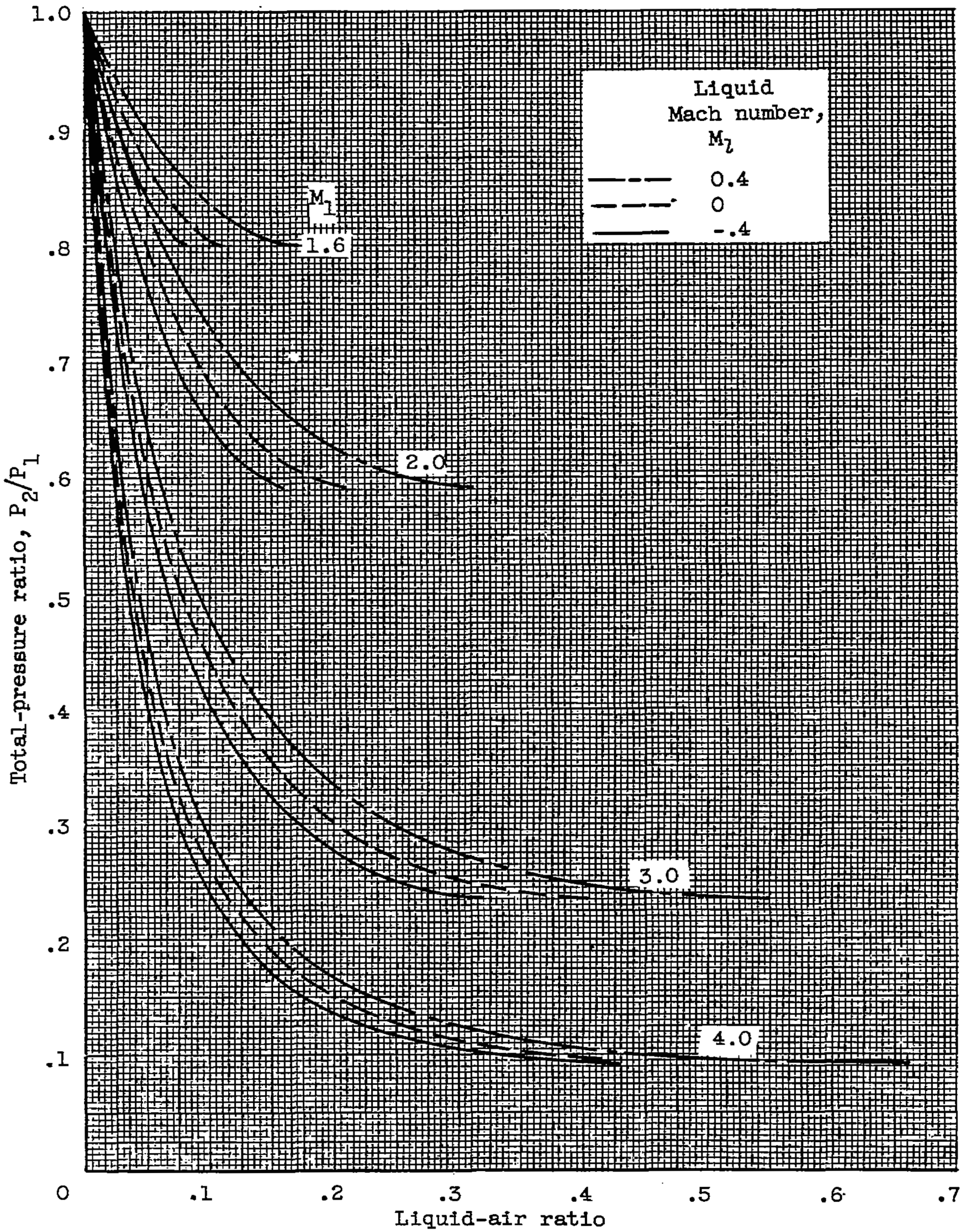

Figure 15. - Total-pressure loss due to fluid injection in a supersonic air stream. $T_2 = T_1$.

NACA - Langley Field, Va.